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STATUS OF ELECTRIC PROPULSION SYSTEMS FOR SPACE MISSIONS

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Introduction

During the past few years, electric propulsion systems have passed from the status of ideas which look interesting but somewhat academic to the status of active research and development, with a firm and growing place in the nation's space program. How has this come about, and is it really justifiable? The electric propulsion cycle itself looks tremendously cumbersome and roundabout — nuclear or solar energy is first converted to heat, which in turn is converted into electrical energy, which is used, either directly or indirectly, to accelerate the propellant rearward to produce thrust. In each portion of this cycle there are inefficiencies and technological difficulties which are far from negligible. Furthermore, electric propulsion systems are so heavy, relative to the thrust that they can produce, that they must be boosted into orbit before they can be used, and they are almost painfully slow in getting in and out of planetary gravitational fields.

In spite of these apparent handicaps, there is increasing realization that of all space propulsion systems that can now be clearly visualized, electric systems show the most promise of minimizing the initial weights and cost of future space missions. The technological problems, although severe, appear to be amenable to solution, and many mission studies have shown that the low accelerations inherent to electric propulsion systems impose no significant trip-time penalties for missions beyond the moon. Consequently, the current interest in, and support of, research and development on electric propulsion systems appears to be fully warranted.

It is the purpose of this paper to summarize in a necessarily brief and incomplete manner, the technological status of electric propulsion systems, and to discuss their current and future space-mission possibilities.

General Performance Considerations

An electric propulsion system can be considered to be made up of two major components — the electric power generator and the thrust generator. The former consists of the basic energy source (nuclear reactor, solar collector, etc.) and the apparatus required to convert the basic energy into electric power. The thrust generator consists of the apparatus which utilizes the electric power to accelerate propellant rearward in the form of a reaction jet. Of the two components, the electric power plant contains by far the largest portion of the total propulsion system weight. This weight is so dominant that no great error is made if the thrust generator weight is assumed to be negligible in comparison.

The thrust generator does, however, have a large indirect effect on the total system weight through the efficiency with which it converts the electric power into jet power. Any inefficiency in this conversion process requires more electric power to produce a given jet power. The electric power generator must, therefore, be larger and heavier, the lower the efficiency of the thrust generator. This relationship is expressed by

$$W_{pp} = \frac{\alpha}{\eta} P_j \tag{1}$$

where W_{pp} is the total propulsion system weight, P_j is the jet power, α is the specific power plant weight (lb/kw produced) and η is the efficiency of conversion from electric power to jet power.

The jet power, in turn, is given by

$$P_j = \tfrac{1}{2}\dot{m}_p v_j^2 = \tfrac{1}{2} F v_j \tag{2}$$

where $\dot{m}_p$ is the propellant ejection rate, v_j is the jet velocity, and F is the thrust $(\dot{m} v_j)$. In terms of specific impulse, I, this relationship becomes

$$P_j = \frac{FI}{45.9} \text{ kw} \tag{3}$$

where F is in pounds and I is in seconds. Combining (1) and (3), and introducing the initial total vehicle weight W_0, yields

$$\frac{W_{pp}}{W_0} = \frac{\alpha I a_0}{45.9\eta} \tag{4}$$

where $a_0 = F/W_0$ is the initial thrust acceleration.

The propellant weight required to complete a mission is

$$\frac{W_p}{W_0} = \frac{Ft}{W_0 I} = \frac{a_0 t}{I} \tag{5}$$

where t is the total propulsion time in seconds. The quantity $a_0 t$ is a measure of the total impulse (thrust × time) needed for the mission and is relatively insensitive to specific impulse.

Equations (4) and (5) show that, although the propellant weight needed for a mission decreases with increasing specific impulse (or jet velocity), the power plant weight increases because the jet power needed goes up. Consequently, an optimum specific impulse exists for each mission, which minimizes the sum of power plant and propellant weight. To determine approximately this optimum specific impulse, the sum of (4) and (5) is differentiated with respect to I. The result is*

$$I_{opt} = \left(\frac{45.9\ (a_0 t)}{a_0\ \alpha/\eta}\right)^{1/2} \tag{6}$$

Substituting this value into the sum of (4) and (5) yields

$$\left(\frac{W_{pp} + W_p}{W_0}\right)_{min} = 2\left(\frac{(a_0 t)\ (a_0\ \alpha/\eta)}{45.9}\right)^{1/2} \tag{7}$$

To illustrate the values of the parameters of interest for space missions, assume that it is desired to have a payload weight about half of the initial weight.

*Much of this derivation, together with a more detailed discussion of the effect of the parameters, is contained elsewhere [1].

Then the weight ratio in (7) must be about 0.5, which means that the parameters $a_0 \alpha/\eta$ and $a_0 t$ are related by

$$a_0 \alpha/\eta \approx \frac{2.9}{(a_0 t)} \tag{8}$$

Trajectory analyses have shown that the parameter $a_0 t$ ranges from about 500 sec for missions such as raising a satellite from a low orbit to the 24-hr orbit to about 5000 sec for some relatively fast round-trip Mars missions. Consequently, for this range of missions

$$6 \cdot 10^{-4} \lesssim \frac{a_0 \alpha}{\eta} \lesssim 6 \cdot 10^{-3} \tag{9}$$

The optimum specific impulse, obtained from (6), (8), and (9), ranges from about 2000 to 20,000 sec. Estimates of the specific weight α (about which more will be said later) range from about 60 lbs/kw for systems which may be available in the next few years (at power levels of the order of 60 kw) down to about 6 lbs/kw for some systems at higher power levels in the more distant future. These values, together with a conversion efficiency, η, give the ranges of initial thrust-weight ratio which must be used to achieve reasonably large mission payloads. For example, with α/η = 60 lbs/kw, initial thrust-weight ratios range from 10^{-5} for round-trip Mars missions up to 10^{-4} for the satellite-raising mission. For α/η = 6 lbs/kw, these values increase to 10^{-4} and 10^{-3}, respectively.

It is apparent from the above discussion and (8) that the effect of high specific power plant weight or low conversion efficiency is to reduce the initial acceleration allowable for a given mission (or given $a_0 t$). This reduction in acceleration increases the time required to perform the mission. For missions to the near planets, for example, it is found that values of a_0 of the order of 10^{-4} are needed to complete the missions in time periods comparable to those of high-thrust vehicles. Consequently, if the specific weight is high (60 lbs/kw range) electric systems are of interest only for missions that require rather moderate total impulse (satellite-raising, or one-way interplanetary probes) or for missions that require long trip times with any propulsion system (probes beyond Mars). For values of α in the 6 lbs/kw range, on the other hand, much more difficult or rapid missions (round-trip interplanetary expeditions or faster deep-space probes) can be undertaken.

The above discussion is considerably oversimplified, in that it applies only for constant-thrust trajectories. Interplanetary missions may, in actuality, require considerable variation in thrust and specific impulse [2, 3]. Nevertheless, the simplified approach provides preliminary information concerning the ranges of propulsion parameters of interest for space missions.

Electric Power Generation

Among the many methods of generating electric power in space, only a few appear promising for the large power, long duration and low specific weight needed for electric propulsion [4]. The basic energy source must be either a nuclear fission reactor or a solar collector to avoid using up excessive mass to generate power.*

To convert the thermal power generated by the nuclear reactor or solar collector into electric power, the turbo-electric method is closest to application. This method is the one most commonly used in ground power stations, and consists in using the basic energy source to heat a working fluid, which drives a turbine-generator system. The basic difference between ground and space power

*Although thermonuclear energy may eventually be very useful for space propulsion, it appears most promising for heating propellant directly rather than for generating power for an electric propulsion system [5] plants is that in the latter the waste heat due to the inefficiency of the thermodynamic cycle can be removed from the working fluid only by radiative heat rejection. Consequently, it is essential to operate the heat-exchange system at the highest possible temperature level so that the radiator size and weight are minimized. This requirement introduces severe materials problems, whose solution is perhaps the most pressing need in the development of lightweight electric power supplies.

Currently, three development programs are underway which utilize the turbo-electric generator approach. The SNAP-2 program, initiated by the Atomic Energy Commission, utilizes a nuclear reactor being developed by the Atomics International Division of North American Aviation. This reactor is to be combined with a turbo-electric system being developed by Thompson Ramo Wooldridge Corporation, and will generate approximately 3 kw of electric power. A solar turbo-electric system, in the same power range, is being developed under the sponsorship of NASA. This system, called Project Sunflower, can utilize essentially the same turbo-electric generator as the SNAP-2, but requires development of a suitable reflector structure to concentrate the solar radiation. Both of these programs were initiated originally to provide long-duration auxiliary power for space vehicles, but they may also be used with small electric thrust generators to provide attitude and orientation control for these vehicles.

The third major development program is a joint NASA-AEC venture, called SNAP-8. This program will provide electric power in the 30–60 kw range specifically for electric propulsion systems. This power range was selected because it is the lowest range which is useful for electric propulsion of space vehicles in the 5000-10,000 lbs range on interplanetary probe or satellite-raising missions. The SNAP-8 program will use a modified version of the SNAP-2 nuclear reactor, together with a turbo-generator system being developed by Aerojet General Corporation under an NASA contract. Initially, the SNAP-8 system is to produce 30 kw of electric power. Running two turbo-electric units from the same reactor will later produce 60 kw.

All three of these active development programs use mercury as the working fluid in a Rankine cycle. In this cycle, the mercury is vaporized and heated to about 900°K by the nuclear reactor. The vapor expands through the turbine, passes through a radiator where the vapor is recondensed, and returns to the reactor to close the cycle. A radiator temperature of about 650°K is appropriate for the mercury cycle at moderate vapor pressures.

Although the precise weight of the three systems is not yet available, approximate estimates are as follows: Sunflower, about 700 lbs; SNAP-2, about 500 lbs; SNAP-8 at 30 kw, about 1000 lbs; and SNAP-8 at 60 kw, about 1600 lbs. These figures are without reactor shielding, which may increase the SNAP-2 weight by about 300–400 lbs, and the SNAP-8 weight by 400–700 lbs. For mission-analysis purposes, even more conservative values have previously been used [6], namely, 2000 lbs for the 30 kw SNAP-8, and 3000 lbs for the 60 kw SNAP-8. Although these values are probably too high, they will nevertheless be retained in this paper to avoid overestimation of the performance of electric propulsion systems.

For power levels in the megawatt range, a number of estimates have show that specific weights of 5-10 lbs/kw may be attainable. In an earlier study [7], for example, a 20,000 kw nuclear turbo-electric system was estimated to weigh about 120,000 lbs, yielding a value of about 6 lbs/kw. At these high power levels, the radiator becomes the heaviest single component of the nuclear-electric system. To reduce this weight, higher temperatures than those attainable with mercury (at moderate pressures) must be used. In this case [7], sodium was selected as the working fluid, operating at about 1400° K turbine inlet temperature and about 1000° K radiator temperature. Use of these temperatures reduces the radiator area to about 1 ft^2/kw of electric power generated.

These estimates in the 5–10 lbs/kw range are based on the assumption that at least two severe technological problems associated with development of nuclear-electric systems will be satisfactorily solved. One of these is concerned with development of materials to withstand the corrosive properties of liquid metals for long periods of time and the other is concerned with radiator damage due to meteorite bombardment. The severity of the materials problem is not yet clear, but it appears that stainless steel may have adequate resistance to corrosion by liquid sodium for temperatures in the vicinity of 1000° K. For the higher temperatures needed in the turbine and other parts of the system, columbium seems promising. The severity of the meteorite penetration problem is also not yet clear, due to lack of adequate data on number, size and velocity distribution of meteorites in interplanetary space, and the uncertainty regarding the mechanisms of penetration. The range of uncertainty given in [7], for a radiator design appropriate for a 20,000 kw system, using 0.025-in. tube wall thickness, is between 2 and 15,000 penetrations during a two-year lifetime. If the high value turns out to be closer to reality, the use of a parallel-tube design, such as that given in [7], is not feasible unless an automatic puncture-sealing process is developed. Merely increasing the tube thickness to reduce penetrations would involve an intolerable weight increase [8]. In the case of manned missions, a remote-control leak detection and spot-welding apparatus might be feasible. A possible alternative approach, involving a tubeless radiator (sheet-metal belt) which picks up heat from a rotating drum, has been suggested [9]. Considerable research on heat transfer between metal surfaces in a vacuum is necessary, however, before the feasibility of such a design is demonstrated. If the scheme is workable, considerable weight reduction, as well as a greatly increased probability of system survival, would result. Consequently, if one is optimistic regarding the development of automatic sealing techniques, tubeless radiators, or the number of penetrating meteorites in space, as well as the finding of corrosion-resistant materials, values of 6 lbs/kw or less for specific weight in the megawatt range appear to be attainable.

Among the possible alternatives to the turbo-generator system, on which most development is now concentrated, are various direct-conversion methods of transforming heat into electric power. Of these, the plasma diode appears to show considerable promise [10]. In this system, nuclear reactor or solar heat is applied, either directly or through a working fluid, to an electron emitter. The electrons tend to stream toward a nearby collector, which is cooled either by direct radiation or by a coolant. Normally, the electron current density is limited by accumulation of space charge to values given by the Langmuir-Childs relation

$$i = 5.56\times10^{-12}\left(\frac{\epsilon}{\mu}\right)^{1/2}\frac{V^{3/2}}{L^2} \tag{10}$$

where i = current density, amps/m^2; ϵ/μ = charge-to-mass ratio of electrons, coulombs/kg; V = potential between plates, volts; and L = distance between plates, meters.

The potential V is essentially the value equivalent to the velocity of electrons leaving the emitter and is typically of the order of 1 volt. Consequently, the distance L must be extremely small to produce significant current density. The introduction of an easily ionized gas, such as cesium, between the emitter and the collector greatly increases the possible current density by neutralizing the electron space charge and reducing the emitter work function.

A possible advantage of a direct-conversion method, such as the plasma diode, is that no rotating machinery is needed. This may result in some weight saving if conversion efficiencies comparable or superior to those of the turbo-generator system are achieved. However, the radiator problems and the high-temperature materials problems are similar, and for the same reactor temperature and radiator temperature, little weight saving, relative to turbo-electric systems, will be expected. If it turns out that the plasma diode can operate for extended periods at temperatures much higher than the turbo-generator systems, a reduction in radiator weight might be possible. Whether the elimination of rotating machinery will increase the reliability of the system depends, of course, on future developments, and cannot as yet be determined. A disadvantage of the direct-conversion methods is the low voltage generated. This requires series connection of many cells to produce the values needed for propulsion.

Electric Thrust Generators

The wide variety of methods possible for producing a propellant jet with electric power can be classified generally into three categories: (1) electrothermal jets, (2) electrostatic jets, and (3) electromagnetic jets.

The first category consists of those devices which use electric power to heat a propellant, which is then ejected through an expansion nozzle such as those used with chemical and nuclear rockets. The principal devices in this category are the electric-arc jets and resistance-heated hydrogen jets (see Figs. 1 and 2). Both devices are limited to fairly low specific impulse for electric propulsion–possibly up to 2000 sec for the electric-arc jet and about 1000 sec for the resistance-heated hydrogen jet. Consequently, their usefulness is limited to those space missions which require relatively small values of the total-impulse parameter, $a_0 t$ (6). These missions include satellite orientation control and raising or lowering satellites between low and high earth orbits.

The electric-arc jet is receiving very wide attention in government laboratories and in industry, not only for propulsion applications, but even more for simulation of conditions encountered during atmospheric entry of missiles and space vehicles. Development contracts for propulsion applications have been awarded by NASA for a 30 kw and a 3 kw arc jet. The former may be used in conjunction with the SNAP-8 power generator and the latter with Sunflower or SNAP-2. Among the technical problems in application of electric-arc jets for propulsion are electrode erosion, nozzle cooling and reducing the percentage of electric power which goes into heating the electrode, nozzle walls and stabilizing resistors. To achieve high efficiency, most of this heat must be recovered by the propellant by means of regenerative cooling. However, if the heat picked up by the propellant from the nozzles, electrodes, etc., becomes a large part of the total heat added to the propellant, the propellant becomes so hot before it reaches the electric arc discharge that its cooling properties have been severely diminished. This means that, for high efficiency, a limit is reached on the specific impulse that can be achieved before it becomes impossible to cool the surfaces with the propellant alone. When this limit is reached, either a radiator must be provided, with a secondary cooling loop, or the surfaces must be operated at sufficiently high temperatures that they produce adequate self-cooling. In either case, the efficiency drops, and the electric-arc jet becomes less desirable for propulsion. The specific impulse limit at which this process becomes dominant depends on future developments in techniques for increasing the effectiveness of heat addition to gases in electric arcs, but is believed to be less than 2000 sec.

The electric-resistance-heated hydrogen jet avoids many of the problems associated with electric-arc jets, in that no electrode erosion or arc heating

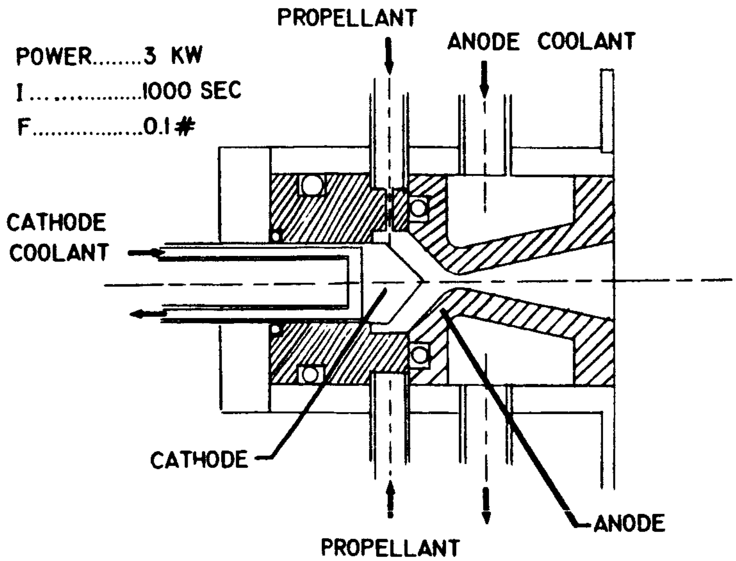

Fig. 1. Arc-heated thrust device.

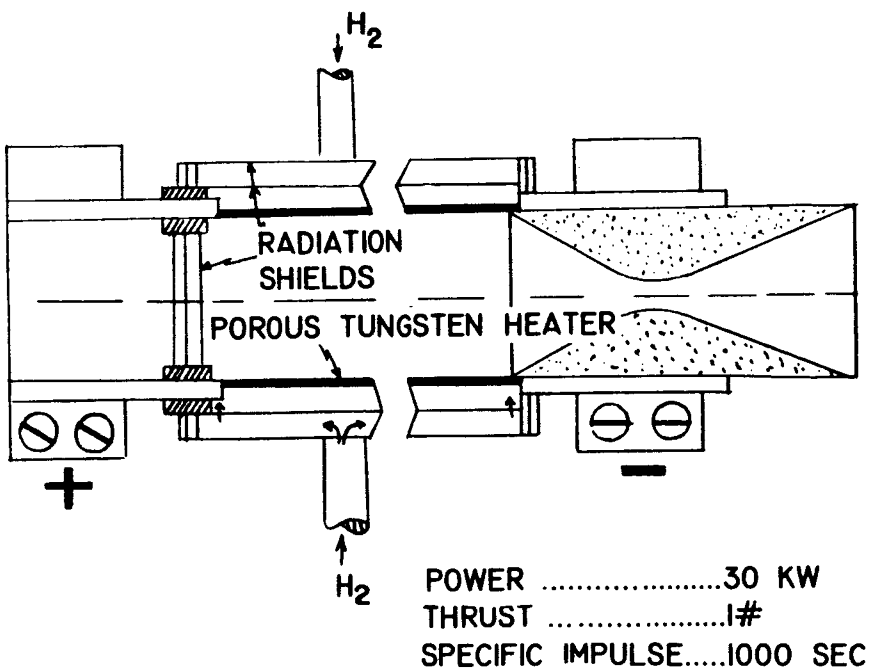

Fig. 2. Resistance-heated thrust device.

of surfaces are involved. The specific impulse is limited, however, to values attainable with hydrogen heated to the maximum temperature that the resistance-heated elements can tolerate. When tungsten in a porous or mesh form is used, this maximum temperature is about 3000°K, which produces a specific impulse of about 1100 sec for hydrogen [6]. As will be shown later, this is adequate for many of the missions for which electrothermal devices are being considered. The resistance-heated jet is therefore a suitable alternative to the arc jet, and may prove superior for many applications.

A research program on this type of electrothermal jet is currently under way at the NASA Lewis Research Center.

Electrostatic Jets

The second category of electrically produced jets consists of those methods in which the propellant is ionized and the resulting positive ions are accelerated rearward with electrostatic fields. Electrons are then fed into the ion beam to neutralize the beam and to avoid charge accumulation on the vehicle.

Considering first the ionization process, among the most promising are contact ionization and electron bombardment. The contact ionization method is being most extensively used at present and consists in bringing easily ionizable substances, usually the alkali metals, into contact with materials that have great affinity for electrons (high work functions), such as tungsten. The particular propellant that shows the most promise of high efficiency of ionization, when used with tungsten, is cesium. To avoid condensation of cesium on the tungsten, and consequent rapid reduction of the work function of the contact surface, the tungsten must be heated to about 1400°K. This heating requirement and the consequent radiation loss constitute the primary nonremovable source of inefficiency in the contact ionization method. This loss is particularly severe for ion beams of low power density, such as those required to attain low specific impulse (below about 6000 sec). As specific impulse increases, the beam power increases, and the radiation loss soon becomes a small percentage of the beam power. At specific impulses higher than 6000 sec, efficiencies of conversion of electric power to jet power greater than 90% are, in principle, attainable.

The electron bombardment method of ionization has the advantage that no heated surfaces are required, and that propellant materials other than alkali metals may be used. This method utilizes a magnetic field to confine electrons emitted in a chamber through which the propellant is fed. The electrons are extracted from the emitter by a small electric field and move along or around the magnetic field lines with sufficient velocity to ionize atoms with which they collide. The resulting ions are then extracted from the chamber with an electrostatic field, as with other ionization methods.

The problems associated with acceleration of the ions, after they are generated, are mainly the following: (1) avoidance of high-velocity ion impingement on electrodes, and (2) attainment of fairly high ion current densities. The first problem involves the durability of the system for the required long operating times and the second involves the size of the beam needed to produce adequate thrust and the efficiency of the accelerator. To attain high current densities, the Langmuir-Childs equation (10) shows that high voltage and/or small spacing between ion source and accelerator are required. Voltage is limited by electrical breakdown, for a given spacing, and the spacing cannot be reduced below a practical minimum determined by fabrication and the need for rugged design. These limitations do not appear to be significant for specific impulses greater than about 5000 sec, where the overall accelerating voltage for the ions is greater than 1000 volts. For this range, a higher accelerating potential can be followed by a decelerating voltage which fixes the final desired specific impulse (or jet velocity). For lower specific impulse, the required overall voltage difference becomes so low that in order to maintain high current density, very large ratios of accelerating voltage to decelerating voltage are required (accel-decel ratios), unless the electrode spacing is very small. Such high accel-decel ratios are likely to produce excessive beam spreading and focusing problems. The electrostatic jet therefore becomes less and less efficient the lower the desired specific impulse, due to the reductions in beam power density relative to power required for ionization. It seems unlikely that efficiencies higher than 50% will be achievable at specific impulses less than 2000 sec with a practical flight accelerator [11].

The final step in an electrostatic thrust device is neutralization of the ion beam by injection of electrons. This neutralization must take place within the beam itself—it is not sufficient merely to eject an equal current of electrons at some other part of the device to maintain overall neutrality of the vehicle. Without beam neutralization, theory shows that the ions, due to mutual repulsion, will quickly diverge, and even return to the neighborhood of the emitter. In laboratory facilities, this ion beam turnaround has been observed only occasionally [12], because of the difficulty of eliminating extraneous electrons. These electrons can originate from a number of sources which would not be available in space. When the ejected ions strike any surfaces in the facility, secondary electrons are emitted which tend to travel back up the beam to produce neutralization [13]. Consequently, most ion beams produced in the laboratory have been more or less neutralized even without deliberate ejection of electrons into the beam. Although some procedures can be devised which minimize extraneous neutralization in ground facilities, it will be difficult to prove adequate neutralization without actual space-flight experiments.

Some of the ion thrust generators currently under investigation at NASA Lewis Research Center are illustrated in Figs. 3, 4 and 5. The first two of these accelerators have fairly large spacing between emitter and accelerator electrodes and are therefore suitable only for quite high specific impulse. Test results for these accelerators were reported earlier [14]. They achieved encouraging values of overall efficiency, η (up to 58%) and, in the case of the reverse-fed accelerator (Fig. 4), undetectably low ion impingement on the accelerator electrode. The accelerator shown in Fig. 5 was designed specifically to produce very high current density with low voltage, such as that needed for low specific impulse. The design concepts are discussed in another paper [11], and feature a pair of fine-wire grids, spaced 1 mm apart, with a scalloped ion-emitter surface to produce beam focusing to avoid ion impingement. Testing is not yet far enough along to discuss experimental performance.

Another accelerator being tested in this program employs an electron bombardment ionization technique in place of the heated-tungsten contact method of the preceding three, and the propellant is mercury instead of cesium. This thrust generator, which was designed by H. Kaufman, has performed very well. Ionization efficiencies of the order of 80% have been achieved with overall conversion efficiencies of 70% at specific impulse of 5500 sec. This efficiency includes the power used for generating the magnetic field.

In addition to the research effort at the Lewis Research Center, NASA has recently awarded a research and development contract for an ion thrust generator capable of producing 0.01 lbs thrust at specific impulses in the 4000-sec range. A similar contract was earlier awarded by the Air Force. These thrust units may eventually be suitable for clustering to produce jet power of the order of 30 kw for eventual use with the SNAP-8 power supply.

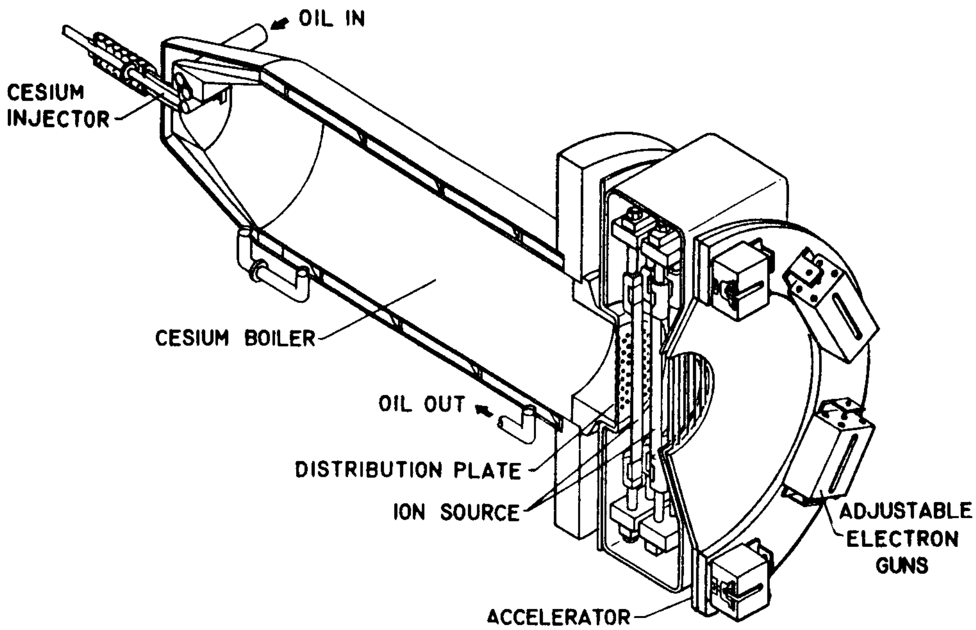

Fig. 3. Through-feed ion engine.

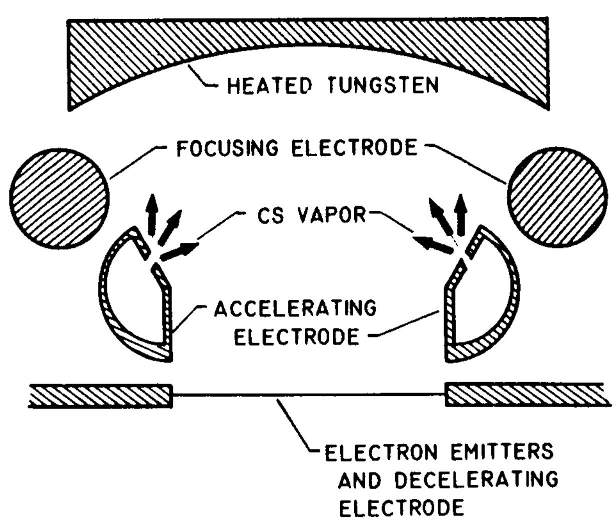

Fig. 4. Ion rocket patterned after high-perveance electron guns.

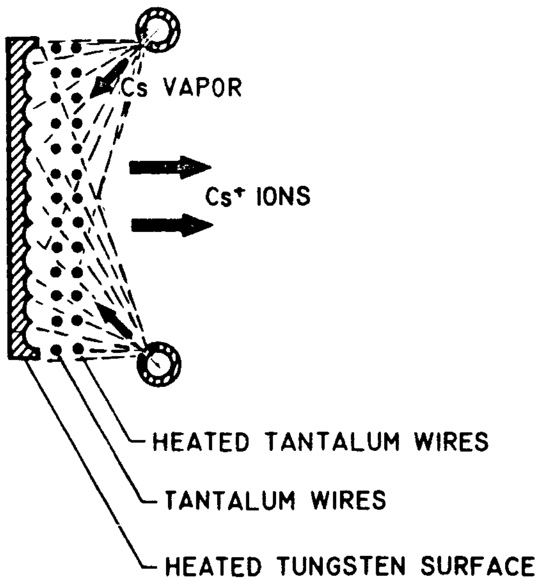

Fig. 5. Ion rocket for high current density. 1 mm spacing between wire grids.

Electromagnetic Jets

The third category of electric thrust generators consists of those devices which accelerate plasma by means of electromagnetic fields. The methods currently being investigated are so numerous and varied that it is difficult to summarize them in a few words. In general, they employ in some manner the basic principle that appropriate combinations of electric and magnetic fields, either stationary, moving or transient, produce forces on a conductor (in this case an ionized gas, or plasma). It is not yet possible to state which of the many proposed methods will eventually be the most successful. To be of interest for space applications, electromagnetic thrust devices must show some superiority over electrothermal jets in the lower specific impulse range or over the electrostatic jets in the high specific impulse range. If it turns out that either or both of the other categories of electric thrust generators encounters insuperable difficulties in achieving either high efficiency, sufficiently high thrust or long lifetime, some electromagnetic devices may well provide the required superiority. Research on these devices is producing valuable insight and data on the behavior of plasmas in electromagnetic fields.

Since the application of electromagnetic plasma accelerators to space propulsion appears to be more remote than electrothermal or electrostatic accelerators, no further discussion will be given herein. Descriptions of many of the proposed methods, together with references to original work, are given in References [1, 4, 6, 8].

Electric Propulsion Research Facilities

The field of electric propulsion research is perhaps most closely related to the field of cryogenics in the design of vacuum testing facilities. Even though some electric rockets produce relatively small thrust, and consequently eject mass at extremely low rates relative to chemical rockets, these mass flow rates are still very large from the standpoint of maintaining the needed high vacuum.

To test electric rockets without appreciable interference from the residual gases in the facility, pressures of the order of 10^{-6} mm Hg or less must be maintained during operation. To accomplish this with reasonable number of diffusion pumps, it is necessary to provide a condenser inside the vacuum tank to condense the propellant emerging from the thrust unit. This condenser is essentially a cryopump. An expression for the amount of surface area required to condense a given mass flow rate was derived by Mickelsen and Childs [15]:

$$S = \frac{2}{3}\frac{\dot{m}v_0}{p} g(f, a) \equiv \frac{2F}{3p} g(f, a) \tag{11}$$

where S is required surface area, $\dot{m}$ is mass-flow rate, v_0 is the initial velocity of the particles to be condensed (jet velocity), p is the pressure and $g(f,a)$ is a specified function of the sticking coefficient, f, and the accommodation coefficient, a. The second expression for S results from the fact that for propulsive jets, $\dot{m}v_0$ is equal to the thrust, F. This derivation is based on several assumptions, which may not be satisfied throughout an actual condenser. Some experiments were conducted by Mickelsen and others at Lewis Research Center to determine the effective value of $g(f,a)$ for several conditions. In one experiment, sodium was injected at thermal velocity into the nitrogen-cooled condenser region of a 5-ft diameter vacuum tank. The function $g(f,a)$ for this case was found to be 0.0435. In another experiment, a small cesium ion beam was injected at velocities of interest for propulsion into a liquid nitrogen condenser having variable surface area. In this case, a value of $g(f,a)$ of 0.14 was obtained. This variation by a factor of three is considered to be the range of uncertainty currently existing for designing condensers for electrostatic propulsion facilities.

For a pressure of 10^{-6} mm Hg, (11) becomes:

$$S = 24 \cdot 10^4 \, Fg(f, a) \tag{12}$$

Using the larger experimental value of $g(f,a)$, this equation shows that a facility designed to test an electric rocket with only 0.01 lb thrust required about 336 ft^2 of condenser area. For flight applications we are interested in thrust values equal to about 10^{-4} times the initial vehicle weight, or about 1 lb of thrust for vehicles launched by the Atlas-Centaur booster. A facility to test such a thrust unit would require about 34,000 ft^2 of condenser surface area.

In addition to the condenser, a test facility must have adequate diffusion-pump capacity to handle leakage and outgassing of noncondensible gases. If liquid nitrogen is the condenser coolant, atmospheric gases and hydrogen or helium are essentially noncondensible gases. If liquid helium were used in all or part of the condenser, a reduction in required diffusion-pump capacity would result, but it is questionable whether such a helium system could be competitive with the diffusion pump on a cost basis.

Current facilities at the Lewis Research Center include four vacuum tanks, three of which are 5-ft in diameter and 16 ft long, and one is $3\frac{1}{2}$ ft in diameter and 7 ft long. Two of the 5-ft tanks have internal nitrogen-cooled condensers with surface area of 730 ft^2. A photograph of this facility is shown in Fig. 6. Most of the tests on cesium-ion rockets discussed previously were performed in these tanks. Pressures in the 10^{-6} mm Hg range have been readily maintained with two 32-in. diffusion pumps in addition to the condenser. As yet, the thrust level of the ion rockets has not been high enough to test the ultimate capacity of these facilities.

To handle thrust units suitable for the SNAP-8 power range, a new facility is currently being constructed at the Lewis Research Center which will have about 50,000 ft^2 of internal nitrogen-cooled condenser area in combination with twenty 32-in. oil diffusion pumps. This tank will be 25 ft in diameter and 80 ft

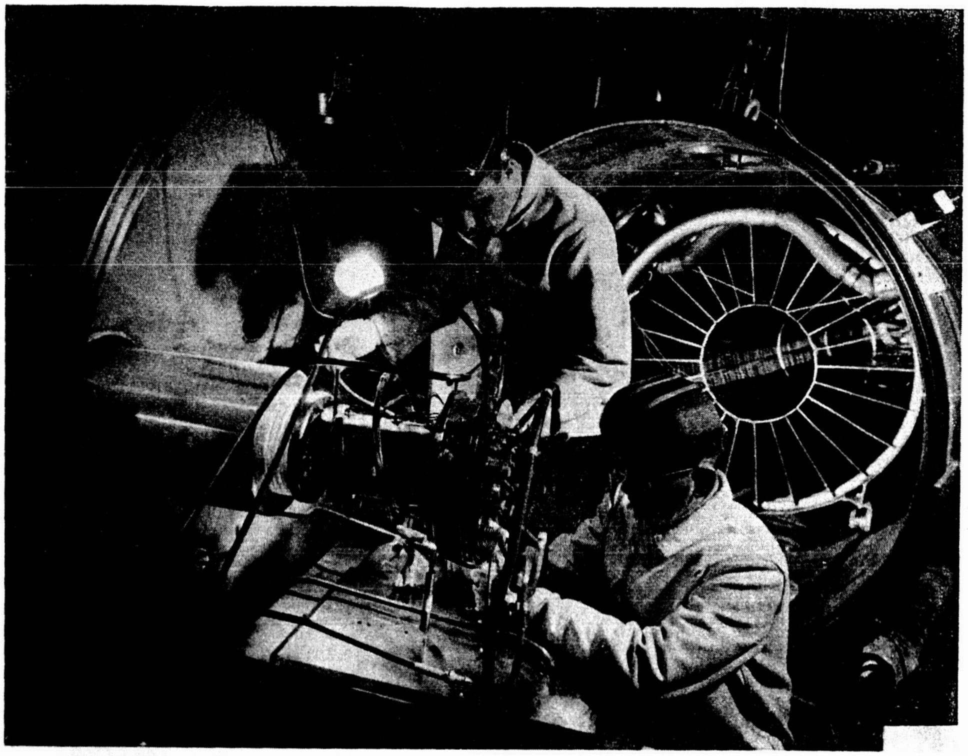

Fig. 6. Electric propulsion research tank number 1. Diameter 5 ft.

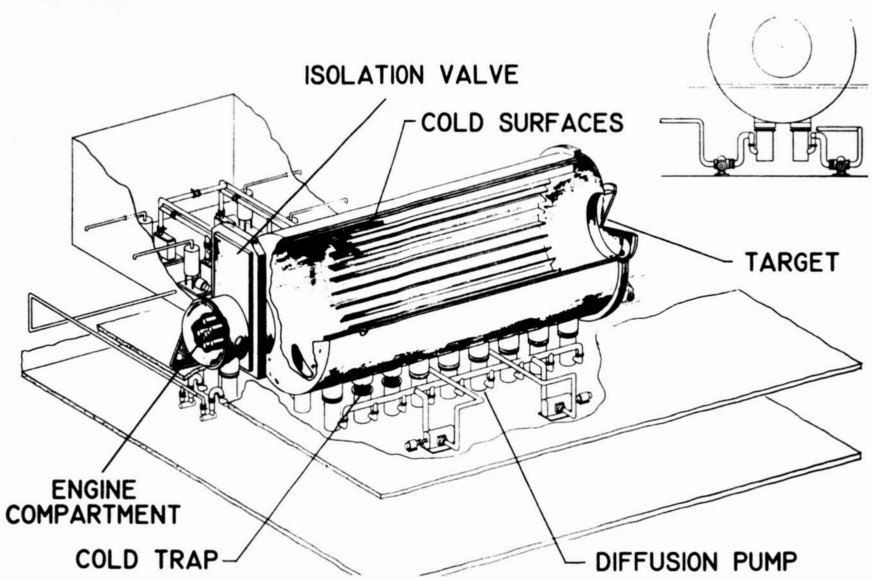

Fig. 7. Sketch of 25-ft-diameter vacuum facility for large-scale electric thrust units.

long to accommodate the condenser area, the diffusion pumps and the contemplated beam sizes. This tank should be adequate for research at thrust levels of the order of a few pounds. An artist's sketch of this tank is shown on Fig. 7.

It is evident that large vacuum facilities are needed to conduct research and development on electric thrust generators in the ranges of thrust of interest for space propulsion. The thrust levels which will be needed for the manned interplanetary missions of the future are of the order of 10–50 lbs and would require a facility with condenser area an order of magnitude greater than that planned for the 25-ft tank. Improvements in cryopumping techniques may eventually reduce these sizes for a given thrust level, or increase the capacity of facilities that will be in existence.

Electric Propulsion Missions

Assuming that the current programs in electric propulsion are successful, and that the required efficiencies and specific weights are attained, what are the advantages of this propulsion system over others such as chemical or nuclear rockets? Using the trajectory theories [2, 3, 16], the initial weights and payloads for a number of missions have been estimated and compared with those needed with other propulsion systems.

24-Hour Satellite Mission

Figure 8 shows a comparison of the payload which can be delivered to the 24-hour orbit (22,600 miles) starting from a low (300 mile) orbit with an initial weight of 9000 lbs. This initial weight is approximately that which can be launched by the Atlas-Centaur vehicle now under development. For the electric system,

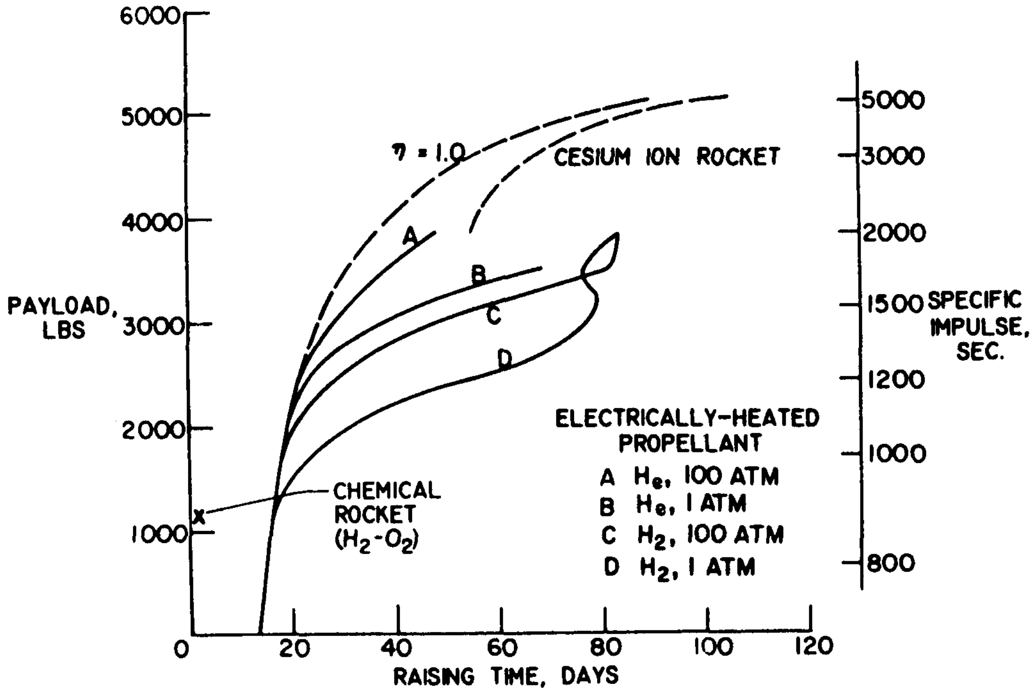

Fig. 8. Estimated payload for raising satellite to 24-hour orbit. Initial weight: 9000 lbs in 300-mile orbit.

use of the SNAP-8 system at 600 kw was assumed. An H_2-O_2 rocket can deliver about 1200 lbs in about 2 days. This payload also must include the electric power supply required for communication. With an electrothermal rocket at a specific impulse of 1000 sec, about 1500 lbs of payload can be placed in the 24-hour orbit in about 20 days, in addition to the 60 kw electric power supply, which was used for propulsion and can now be used for communication. Higher weights can be delivered if higher specific impulses are used, but the raising time is correspondingly longer. It appears, therefore, that an electrothermal thrust generator, combined with a 60 kw electric power supply, should be very useful for the satellite-raising mission.

Interplanetary Scientific Probes

Electric propulsion systems are even more advantageous for unmanned scientific deep-space probes, where it is desirable to have large amounts of electric power available for transmission of information from large distances. Shown in Fig. 9 are the payloads that can be delivered into a low orbit around Mars, starting with an initial weight of 9000 lbs in a 300 mile earth orbit. The payload is shown as function of total trip time for electric rockets powered with 30 and 60 kw SNAP-8 power supplies and for a nuclear rocket with specific impulse of 800 sec. The power plant weight for the nuclear rocket was assumed to be 2000 lbs, which may be a feasible minimum weight, although it appears that no such rocket is currently being planned. Curves are shown for the net payload after subtraction of the electric power plant weights indicated. Using atmospheric braking at Mars to save propellant, the nuclear rocket can carry about 1500 lbs in addition to a 30 kw electric power supply. An electric rocket, with a total trip time of about 240 days, can carry about 2000 lbs of payload to Mars, in addition to the 60 kw power supply. Perhaps a more valid comparison would be with a chemical rocket rather than with a hypothetical nuclear rocket; such a comparison, of course, is even more favorable to the electric rocket. A booster the size of the Saturn vehicle ($1\frac{1}{2}$ million lbs thrust) would be required to carry as much payload and electric power supply to Mars as the electric rocket with the Atlas-Centaur booster (about 400,000 lbs thrust).

Manned Interplanetary Missions

A comparison was made [3] of the initial weight that must be launched into a near-earth orbit to undertake a fairly elaborate 8-man expedition to Mars and return. The trajectories used for the electric propulsion mission were members of a simple family, which permitted consideration of indirect paths. These indirect paths were found to reduce greatly the total round-trip time for electrically propelled interplanetary missions, just as indirect coasting trajectories have been found to reduce round-trip time for high-thrust chemical or nuclear rockets. Results are shown in Fig. 10 in the form of total initial weight in a 300 mile earth orbit as function of total round-trip time. The curves for the nuclear rocket are based on optimum trajectories described by Dugan [17]. The two curves for the nuclear rocket result from the fact that different families of trajectories are found to be optimum in the two ranges of total trip time.

It appears that, if specific weights of the order of 10 lbs/kw or less are achieved, electric rockets can complete a manned round-trip mission faster than nuclear rockets, at least for the lower range of initial weights. Thus, for a 600 day trip, the initial weight in orbit would be about 300,000 to 600,000 lbs with electric systems having values of α in the 5 to 10 lbs/kw range, and about 1,200,000 lbs for a nuclear rocket with a specific impulse of 1000 sec. Since the trajectories assumed for the electric rocket were not optimum, it is possible that further trajectory research will show that even faster trips are possible for given initial weights.

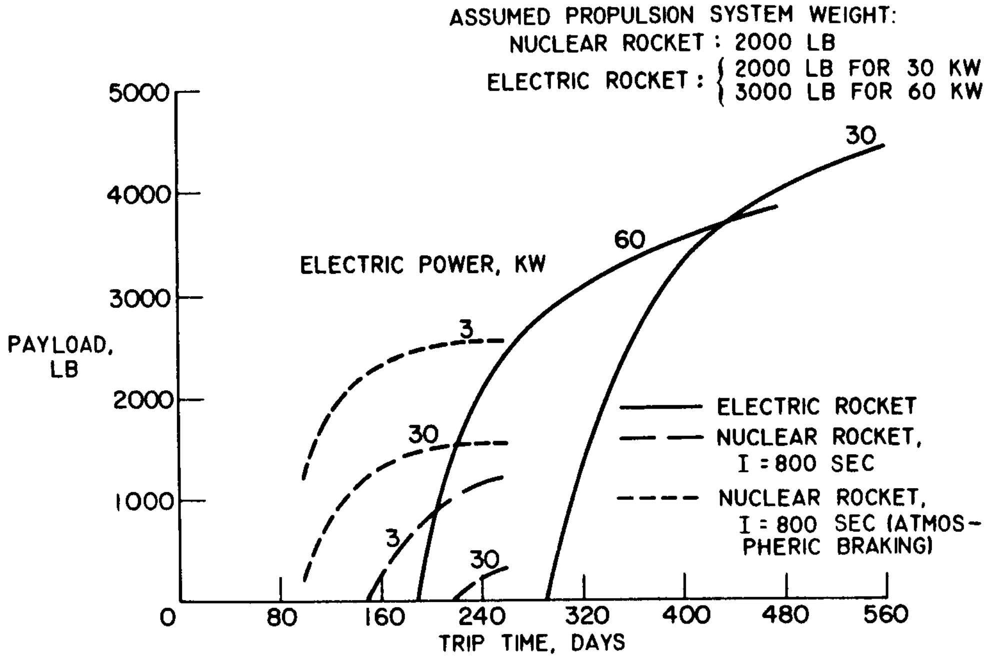

Fig. 9. Payload weight delivered to Mars. Initial weight: 9000 lbs in 300-mile orbit.

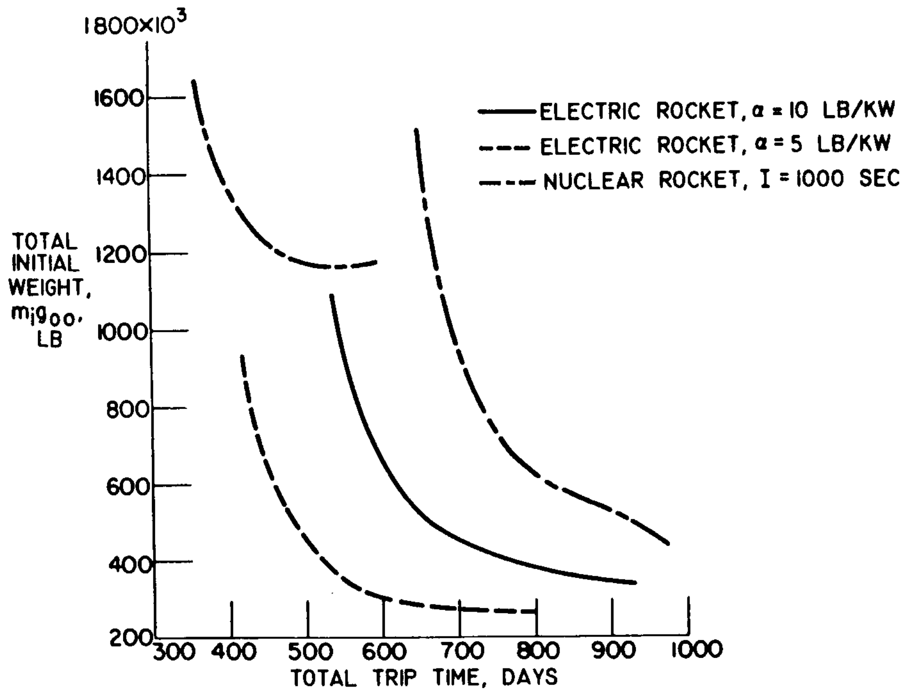

Fig. 10. Initial weights in orbit for 8-man Mars mission.

It can be concluded that electric rockets, despite their low thrust-weight ratio, are capable of accomplishing interplanetary missions in times comparable with those of high-thrust vehicles and with considerably less initial weight. Missions such as those shown in Fig. 8 are, of course, still at least a decade or two in the future. Many concepts of what is feasible will undoubtedly change in that period of time. It appears clear, however, that electric propulsion systems will perform a vital role in the unmanned and manned space explorations of the future.

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